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The MIT Lincoln Laboratory Pulsed Plasma Thruster

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A FINAL REPORT ON THE LES-8/9 PULSED PLASMA THRUSTER*

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Abstract

This paper in conjunction with a previous publication forms a final report[1] on the LES-8/9 pulsed plasma thruster. It briefly mentions the research phase of the M.I.T. Lincoln Laboratory pulsed plasma thruster (PPT) program and presents some important findings. It discusses the development program for the LES-8/9 flight PPTs, in particular: 1) the testing of PPT electronic components, 2) the changes made in the electronic circuitry as a consequence of thruster operation, 3) the testing of flight PPTs, and 4) the integration of PPTs to the LES-8/9 satellite. As a result of a series of problems in the PPT electronics the thrusters were eventually replaced on LES-8/9 by a cold ammonia gas system. Work continued, however, and ultimately the PPT system was proven flight worthy by successfully completing the thruster ground test program.

I. Introduction

Lincoln Laboratory's experience with pulsed plasma propulsion began in 1968 with the flight of four PPTs aboard the LES-6 communications satellite[2]. From then until 1973 a research program into the plasma and electrical characteristics of PPTs was conducted[3-9]. The intent was to gain confidence in PPT reliability and if possible improve performance. Two of the major results of those investigations were: 1) a thruster design with a vastly improved thrust to power ratio[6], and 2) a thruster model which presented a correlation between geometrical and electrical circuit parameters and thrust performance[8]. The improvement in thrust to power was accomplished by feeding the fuel along the discharge path (side feed) rather than behind it. This concept was picked up at Fairchild Hiller where it was developed into an efficient mlb PPT[10,11].

The research phase led to the beginning of work on a flight system for LES-8/9 in 1973. Ten thrusters were made, six of which were to fly. They were to provide three-axis attitude control and orbit control on a 1000-lb satellite[1]. A photograph of one of the LES-8/9 flight thrusters is shown in Fig. 1. It weighs 15.9-lb including fuel and electronics. A 17-μf capacitor (at the bottom) is charged to 1530-V. When a spark plug in one of the nozzles is ignited, the capacitor which is connected through a stripline to cathode and anode electrodes in the nozzle is discharged across the face of a solid teflon fuel bar. The resulting plasma produces an impulse bit of 67 μlb-s at a 1000s specific impulse. Each thruster carries enough fuel for 1645 lb-s total impulse. The nozzles and thrust vectors are canted 30° to the center line. The 30° feed system was developed at Fairchild Hiller[4] for three-axis attitude control[1]. The thruster's power conditioning, logic and spark plug circuitry are located in the two boxes behind the capacitor.

From 1973 thru 1974 prototype PPTs were life tested. The thruster consisted of the energy storage capacitor, stripline, discharge chambers and fuel feed system. In those tests the breadboard electronics were located outside the vacuum chamber for easy access in case of electrical difficulty. Our prime concern was evaluating the basic thruster. It could not be exposed to air once a test started. During the test period three prototype PPTs operated successfully for a total of 6900 lb-s. Each met or surpassed the satellite mission requirement of 1250 lb-s and each was qual or flight vibration and shock tested prior to life test.

These tests proved that the thruster design itself was reliable but revealed some electrical problems with the discharge initiating (DI) circuit energy storage capacitors and SCRs. More problems were encountered when the power conditioning electronics was installed on a thruster and operated in vacuum. Paschen breakdown, electromagnetic interference, electrostatic pickup and plasma ground currents all contributed to failure at one time or another of the PPT electronics.

Eventually the electrical problems were solved and six thrusters with power conditioning electronics were installed on LES-8/9 for a series of satellite-PPT integration tests in air and vacuum. Air operation was accomplished with ignitrons and was found to have no effect on the other satellite systems. The problems that were encountered during vacuum operation were due in part to test setups and in part to improper shielding against electromagnetic radiation and plasma particles. After the proper changes were made the integration tests were successfully concluded.

At this point we needed more confidence in the long-term operation of the PPT system. Two tests were conducted. They were: 1) a life test in which a flight PPT was successfully operated in vacuum for the total impulse of the LES-8/9 mission, and 2) an abbreviated life test (1/10 lifetime) in which three flight thrusters were operated in a spacecraft configuration. This test ended in failure when certain electrical components in the power conditioning circuits failed. Because of the launch schedule there was no time to modify and retest the PPTs, so the decision was made to replace the LES-8/9 PPTs with a cold ammonia gas system. However, to prove that the PPTs are a viable flight thruster circuit modifications were made and the test was again conducted. This time it was successful.

II. Discharge Initiating Circuit Capacitors and SCRs

The discharge initiating circuit consists of two paralleled 1-μf capacitors in series with a silicon controlled rectifier (SCR) and the primary of a 3:1 step-up pulse transformer. The spark

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plug is in the secondary. When the SCR switch is closed the capacitors which have been charged to 650-V are discharged through the primary of the transformer. When the voltage reaches ∿1000 volts across the spark plug it ignites and the engine fires.

We initially procured the same capacitors and SCRs used in previous programs because they had been successful. However, during the life tests of prototype thrusters one or two 1-μf capacitors and SCRs failed. It became necessary to learn more about these devices.

After discussions with the manufacturer and examining shorted capacitors, we concluded that they were not designed to handle the peak currents of our DI circuit. We improved the design by doubling the metal thickness of the metalized paper winding and by increasing the area of the solder contact between the current leads and the extended foil.

These capacitors were made by Sprague (1-μf, 1000-V, Model No. 118P1021). We screened them as follows: Two 1-μf capacitors connected in parallel were discharged 300,000 times into a 1-Ω, 1-μh load through a rotating mechanical switch. This produced a peak current per capacitor of 170-A at 10% voltage reversal as compared to 140A peak and 0% reversal in thruster operation. A capacitor was accepted if the increase in dissipation factor was <200% and the decrease in capacitance was <10% of the original values. Eight capacitors which passed screening were then put on life test. Four were operated at 1000 volts at room temperature and the other four at 650 volts and 75°C. The tests were terminated after 21 million discharges. They had surpassed the LES-8/9 mission requirement of 18 million discharges. All capacitors were still operational.

The SCRs shorted as a result of a leakage current buildup in the RTV compound used to encapsulate the SCRs. These SCRs were designed for AC operation and the manufacturer could not guarantee an acceptable leak rate for our DC bias condition. We switched to glass passivated SCRs which do not have this problem to the same extent.

We chose the Motorola MCR 1718-10 SCR, rated at 800-VDC, and capable of handling pulse currents to 1000-A and di/dt to 1000-A/μs. We bought 154 units and screened 24 for flight according to the following test sequence. Leakage current was monitored while each SCR was: 1) thermally stressed from -60°C to +110°C at 650-VDC three times, 2) biased at 650-VDC for 13.3-hr. at 25°C, 33.3-hr. at -35°C and 88.8-hr. at 75°C, and 3) reversed biased at -650 VDC for a minute. Typical room temperature leakage currents were a few μa. An SCR was rejected if its leakage current increased significantly. Those accepted for flight had no change. Next, each SCR was operated in a thruster discharge initiating circuit for: 1) 30,000 shots at 25°C, 2) 70,000 shots at -35°C and 3) 200,000 shots at 75°C. Those that did not degrade were helium leak tested and those with leak rates $<10^{-9}$ cc/s were accepted. They were potted in EpoTek H72. This epoxy was used primarily to provide mechanical strength for the cathode and gate terminals at the glass interface with the can. Without it gross leaks develop at the base of the terminals when the terminals are bent or twisted. The epoxy also has excellent sealing properties.

III. Electrical Problems Caused by the Arc Discharge

When we first turned on a thruster in vacuum with its power conditioning electronics, it lasted exactly one shot. Many of its electrical components failed. This was the beginning of a series of electrical problems that were encountered during the integration of the electronics and the basic PPT. These were due in large part to the proximity of the electronics package to the 15,000-A arc discharge. The effect of the arc discharge and plasma environment on operation of the logic and DI circuitry had not been fully appreciated during the design phase. The following is a summary of the circuit modifications made as a result of thruster operation.

During a thruster discharge the vacuum surrounding the thruster is filled with a conducting plasma which parallels the arc discharge. It runs from the anode electrode to the tank walls, from there to the ground shield of the line feedthroughs, along the cables to the logic input and back to the cathode electrode. When the first thruster was turned on a 200-A current spike flowed through that path into the ground of the power conditioning control logic box and destroyed IC components. The fix was to install a common mode choke across the plus and minus charging lines from the power converter to the 17-μf energy storage capacitor. The choke presents a low impedance for currents that flow both ways through it, i.e., the charging current, but a high impedance for the one-way currents due to the plasma. The choke reduced plasma return currents to a few amps. It should be mentioned that plasma return currents do not exist in space unless the plasma contacts a grounded portion of the spacecraft which is common with the thruster ground. Vacuum testing is therefore a worst case condition for plasma return currents.

Noise (probably due to electromagnetic radiation) during the main arc discharge caused the spark plug sequencing logic to go awry at times. Occasionally two plugs in different nozzles sparked simultaneously. At other times the wrong plug would go off.

The simultaneous firing was due to the main arc discharge generating a noise spike which triggered an SCR in the other nozzle. At that time there were separate DI capacitors for each nozzle. The problem was simply eliminated by removing one set of 2-μf capacitors and operating the plugs in both nozzles from the same set. This points out that in general it is best to design PPT circuitry so that sections not being used are in an "uncharged" or quiescent state during the main discharge.

Several modifications were made to the control logic itself because of the noisy environment. The change required to correct the plug sequencing problem is an example. The arc discharge occasionally scrambled the D flipflop which alternated spark plug firings. The fix was to add a feedback network which slowed up the circuit and prevented it from responding to noise spikes which were appearing on the clock.

"Latch-up" is another noise induced problem. It occurs when an SCR receives a spurious trigger signal during the changing cycle. Capacitor charging works as follows: When the voltage reaches 1530-V on the energy storage capacitor, a trigger command is generated which turns off the converter and fires the thruster. The 650-V for the D1 circuit is tapped from the 1530-V power transformer. The SCRs are in parallel with the 650-V. When latch-up occurs, this portion of the transformer is short circuited and the converter is incapable of charging to 1530-V. It never turns off. The anti-lockup circuit is simply a timer which automatically shuts off the converter after a preset time and allows the SCR to come out of conduction.

During a flight thruster life test we observed self triggering of SCRs. Pick-up from the main discharge produced a dV/dt of ~600-V/μs across the anode-cathode leads of the SCRs not being commanded. This was occasionally sufficient to self trigger an SCR and dump a portion of the energy left in the 2-μf capacitor. This dV/dt triggering was eliminated by placing a 33-Ω resistor across the gate-cathode leads. This reduced the CdV/dt current in the SCR gate, where C is the junction capacitance, and increased the dV/dt capability of the device. After 33-Ω was added the spurious triggering was virtually eliminated. See "Flight Thruster Life Test". As a result of this experience, a high dV/dt trigger level was used as one of the screening criteria for choosing flight SCRs with 33-Ω added to the circuit. With properly screened SCRs, dV/dt triggering was completely eliminated.

The above problems lead us to the following design philosophy. First, work very hard to make the circuit operate properly under worst case conditions by filtering, shielding, etc., and second, assume that the circuits will occasionally malfunction in spite of your best efforts and design out any serious consequences. The "latch-up" fix mentioned above is a good example.

IV. Satellite Integration

The LES-8/9 satellites provide up and down communication links at both UHF and K-band and crosslink at K-band. An S-band subsystem provides for the transmission of telemetered data from LES-8/9 to earth terminals. In addition there are other systems which contain sensitive amplifiers and logic circuitry that might be effected by thruster operation. For this reason the thrusters (six) were installed on a LES-8/9 satellite for a series of integration tests performed first in air, for ease of troubleshooting, and then in a vacuum tank.

An ignitron load was designed for air operation. A General Electric ignitron (#GL-37248) was connected to the anode and cathode electrodes of each PPT through a low inductance, low resistance path. During operation a fire signal to any one of the four spark plugs was diverted to trigger a thyratron which was across a 30-μf capacitor charged to 500-V. This discharge triggered the ignitron which in turn discharged the thruster. For the air tests to be meaningful the peak discharge current had to be comparable to that for vacuum operation. The ignitron design effort was directed towards duplicating the low inductance and resistance of a vacuum arc discharge (which was 60-nH and 50-mΩ, respectively). We achieved a 12,000-A peak current with the ignitron as compared to 15,000-A peak in vacuum.

The air tests were successful. No other satellite system malfunctioned as a result of thruster operation and in retrospect this is understandable. Thomassen at Lincoln Laboratory[9] and Palumbo, Begun and Guman at Fairchild Hiller[11] found that the source of electromagnetic noise from PPTs is the deceleration of the high energy electrons which initiate the arc discharge. They collide with the plasma and anode and generate a white noise pulse 200 to 300-ns in duration. Since this mechanism does not exist in ignitron operation the major source of EM noise is not present. The other major source of interference, plasma ground current, obviously did not exist either. Air tests are useful therefore for checking out the thruster command systems, but not for testing noise susceptibility.

Next, a series of satellite integration tests were performed in vacuum. PPTs were integrated with satellite structure and housekeeping systems. No RF systems were included. The thrusters were fired in different sequences and at different rates to simulate typical stationkeeping, station changing and attitude control operations. Figure 2 is a time-elapsed photograph of all three thrusters on one face firing sequentially in a station change maneuver. Both the telemetry system and the infrared sensor system were adversely affected by thruster operation.

The telemetry problem was due to the test setup. When the PPT plasma exhaust contacted the chamber walls current returned to the spacecraft by way of the ground shield on the telemetry test cable. The L/di/dt drop along the cable ($di/dt \approx 2A/2\mu s = 10^6 A/s$) was sufficient to produce several voltage spikes between the satellite and the tank ground. This produced errors in the telemetry output data. Common mode rejection and filtering modifications to the cable eliminated the problem.

The infrared sensor was affected by EM radiation and electrostatic pick up other than plasma return currents. Here the need for thorough shielding was emphasized. The shield on the cable to the infrared bolometer stopped several inches short of the box. Electromagnetic pickup on the exposed cable, which was about two feet from the nearest thruster, interfered with logic circuits. Another cause of the problem was plasma leakage. The thrusters were installed on the satellite without spark plug covers. As a result plasma blew backwards through the plug openings during a discharge onto the satellite harness and disrupted logic operations. After extending the shield to the bolometer box and installing the spark plug covers, the IR sensor system operated without any further difficulties.

The lesson here is that all electronic boxes and cables should be adequately shielded against electromagnetic radiation and electrostatic pickup. The number of test cables from the tank to the satellite should be kept to a minimum. Each cable can generate unwanted noise from electrostatic pickup and plasma current L/di/dt drops.

There was also some interference between PPTs (cross firing). Occasionally an adjacent thruster discharged simultaneously with the one being commanded. This did not happen between thrusters on opposite sides of the satellite. An examination of the thruster command system indicated that it was operating properly. A series of experiments was then initiated to determine the cause with the following results: When a thruster discharges, a portion of the plasma spills into the nozzle of an adjacent thruster. If that thruster is charged and the vacuum tank pressure is $>10^{-5}$ mm-Hg, the plasma will break down the electrode gap and discharge the thruster. At pressures $<10^{-5}$ mm the event rarely happens and when horns (i.e., plasma deflectors) are added, it is eliminated altogether. As a result of these findings, horns were added to all thrusters (see Fig. 5) and tests of multiple thrusters were performed at pressures $<10^{-5}$ mm with no further crossfiring.

V. Flight Thruster Life Test

As mentioned earlier, we had satisfactorily tested the PPT without electronics and knew it was reliable. However, we did not have any long-term operational data on a complete thruster, i.e., a thruster with power conditioning electronics. As soon as the first flight thruster was complete (see Fig. 1) it was put through a qualification vibration and shock test and then put on life test.

Figure 3 shows the test arrangement in the vacuum tank. The test was designed to reproduce the thermal environment of the satellite and space operating conditions. The PPT was attached to a portion of a LES-8/9 decagon structure which contained heater resistors to represent the heat dissipation of surrounding electronics. The nozzles with horns attached protruded through a thermal blanket as shown in Fig. 3.

The test began with our original epoxy encapsulated SCRs. We had not yet completed our screen of Motorola flight SCRs for inclusion in this test. This was fortunate, because after 180,000 shots one SCR shorted out. Subsequent examination showed that all four had gross air leaks and the shorted one had arced internally when the pressure inside was reduced to the Paschen breakdown region. As a result of this experience, we included a leak test in our SCR screening procedure and sealed all flight candidates with EpoTek H72. The four bad SCRs were replaced and the test was started over again.

It was performed under continuous vacuum for six months, 24-hrs. per day at the flight firing rates of 1 and 2-pps. The initial portion of the test was performed with the PPT exposed to one solar constant. During another period the thruster was thermally cycled from -15°C to +55°C, five times. The test was terminated when the LES-8/9 total impulse requirement (1250 lb-s, 18.5 million shots) was met.

Fuel consumption was monitored throughout the test and the mass-per-shot variation for the left nozzle is shown in Fig. 4. The results are similar for the right nozzle and confirm an earlier prediction[1] that the variation would be from ~25 to 35 μg/shot.

As mentioned in Section III, dV/dt pickup can cause SCRs to self-trigger and their spark plugs to discharge. The fix had not been incorporated when this test was run. The misfire rate for the four plugs was 1 to 23%. We ran the entire test this way and periodically recorded the spurious spark rate. The rates remained constant throughout the test and were as follows per 1000 thruster discharges: plug 1, 74 sparks; plug 2, 9 sparks; plug 3, 0 sparks and plug 4, 228 sparks. After insertion of 33-Ω across the SCR anode-cathode leads dV/dt triggering disappeared except in plug 4 where it was reduced from 228 to <3 sparks per 1000 discharges. Fuel consumption was not effected as was evident from the fuel measurements during the test and the evenly eroded teflon faces at the conclusion of the test. We concluded that dV/dt triggering if it occurred would not seriously affect PPT operation.

VI. Test of 3 PPTs

It was felt that we needed more confidence in the electronics and operation of the thrusters in a spacecraft configuration. At this point in time we only had 6 million shots on the flight thruster life test, dV/dt triggering had been observed and plasma spill-over had affected thruster operation during satellite integration tests.

To test all the modifications made to date, we decided to test three thrusters in the spacecraft configuration shown in Fig. 5 at 2-pps until each thruster had accumulated 2 million discharges. At that time the thrusters would be thermally cycled between -15°C and 60°C three times.

The thrusters ran flawlessly for 2 million discharges. Flight Motorola SCRs with 33-Ω shunts were used and no dV/dt triggering occurred. Operation in the $<10^{-5}$ mm range and the use of horns eliminated the plasma cross-fire problems which occurred during the satellite tests.

It looked like the test was going to be successful until the last day when thermal cycling began. At -15°C a nozzle in one of the thrusters stopped firing and at +55°C the capacitor charge time in another thruster became too long for operation at 2-pps.

The test was terminated and it was decided to replace the LES-8/9 PPTs with a cold ammonia gas system. We had not been able to reach a plateau of consistently successful test results to instill confidence in the PPTs as a flight system. Although the problems were considered solvable, they were not within the time span available. After the cause for the failures could be found and fixed, the test was to be repeated and the thruster program brought to an end.

VII. Successful Repeat of 3-PPT Test

The thruster logic was designed to accept four command inputs, left and right auto and left and right manual. Left and right refer to the left and right nozzles and auto and manual refer to a command mode of operation. All four inputs are logically combined in a 4-input NAND gate and two of the inputs drive other gates used in decoding right/left and auto/manual. This input gate in one of the thrusters had developed a high resistance leakage between its inputs. The wrong inputs could then cross couple out to influence the gates sensing right/left or auto/manual. At room temperature the logic threshold was below the cross coupled signal but as the temperature was lowered to -15°C the logic threshold increased enough to cause firing the wrong nozzle. This same gate was deteriorated in another thruster but not to the point of affecting operation.

We conclude, then, that these input gates were bad before the start of thermal cycling. In fact since we did not check them prior to this test, they could have been bad when the test started. Deterioration may have occurred during earlier tests before command inputs were shielded from the plasma.

The following changes were made: The left and right manual inputs were removed to minimize the number of external wires connected to the electronics and to minimize the number of problem areas. Thruster operation is identical in both the auto and manual modes so removing the manual mode does not affect the thruster. The four-input NAND gate was no longer necessary and was removed. In addition a 510-Ω resistor, 1-μf capacitor filter were added to the left and right auto inputs (.5 ms time constant). We thought this would be effective since events associated with thruster operation last for only 10s of μs.

The problem of the capacitor charge time becoming too long for operation of 2-pps was caused by a power transformer which had arced internally and loaded down the power converter. We believe that this transformer was a prototype unit, one without adequate insulation, and never should have been installed.

After the aforementioned logic changes were made and a flight power transformer installed the 3-PPT test was repeated. The "health" of the input gates was monitored daily by measuring the open circuit voltage at various external command inputs. Overall health of the converters (and transformers) was also monitored daily by observing the energy storage capacitor charge times. As before we ran at 2-pps until the last day of the test at which time we thermally cycled the thruster from -15°C to +60°C. All thrusters operated perfectly and the test was terminated. A curve tracer check of the input gates confirmed that they were still good.

VIII. Summary and Conclusions

This paper discusses the tests and screening procedures used for critical electronic components, suggests precautions to be taken during satellite integration tests and describes the effects of and design changes dictated by EM radiation, electrostatic pickup and plasma currents on thruster electronics.

Although the thrusters never flew on LES-8/9 because of electronics problems the PPT system was eventually proven flight worthy by the following series of tests:

1. A successful LES-8/9 life test (1250 lb-s) of a flight thruster after it passed qualification vibration and shock.

2. A successful long-term test of three flight thrusters in a LES-8/9 configuration.

3. A successful integration to the LES-8/9 communications satellite.

With proper design we were able to prevent EM radiation and plasma charge from interfering with the PPT circuit operation. However, next time I would recommend that the electronics be located in a separate package as far from the propulsion system as possible.

IX. Acknowledgements

The author would like to thank the following people for their contributions to the Lincoln Laboratory PPT program. Keith Thomassen and Albert Solbes, consultants from M.I.T., provided many of the ideas and experiments which were the basis for our research program. Greg Fuller, an electromechanical technician, was with the program from the start. As a mechanical designer and machinist he was invaluable during our research phase and again during flight design and fabrication. He designed portions of the thruster and was responsible for flight assembly. I would also like to thank Dick Sullivan, Staff Specialist. Although he was with the program for only a short while, he had the impact of three people over twice the time span. And of course, the Group Leaders, Bill Sarles and Frank Floyd, and the LES-8/9 Project Leader, Don MacLellan, who stuck with the PPTs to the last possible moment.

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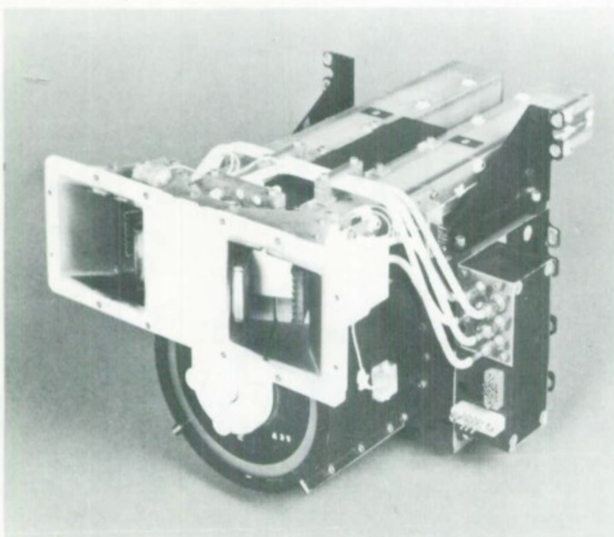

Fig. 1. LES-8/9 Pulsed Plasma Thruster

Fig. 2. Three PPTs firing on LES-9

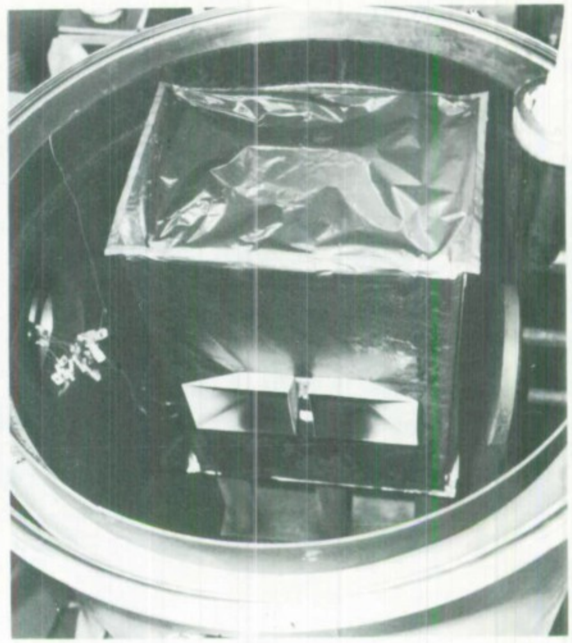

Fig. 3. Thruster life test setup in the vacuum chamber. The thruster is attached to a decagon structure inside a thermal blanket enclosure. The nozzle horns are seen protruding through the hlanket.

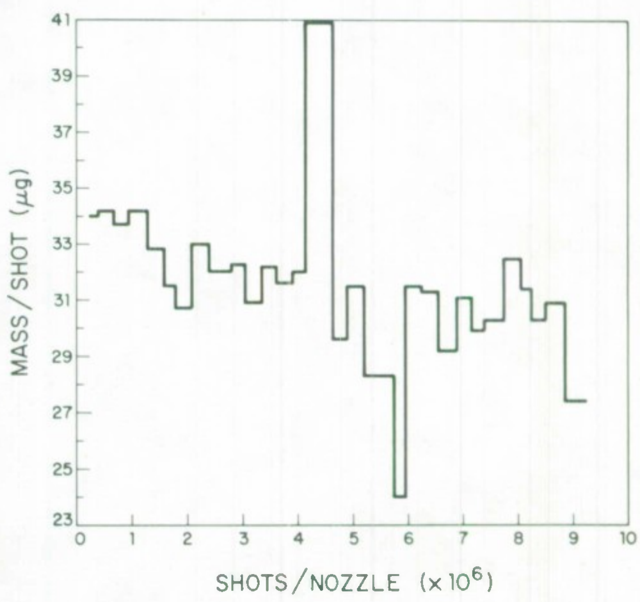

Fig. 4. Fuel Consumption in the Left Nozzle during Flight Thruster Life Test

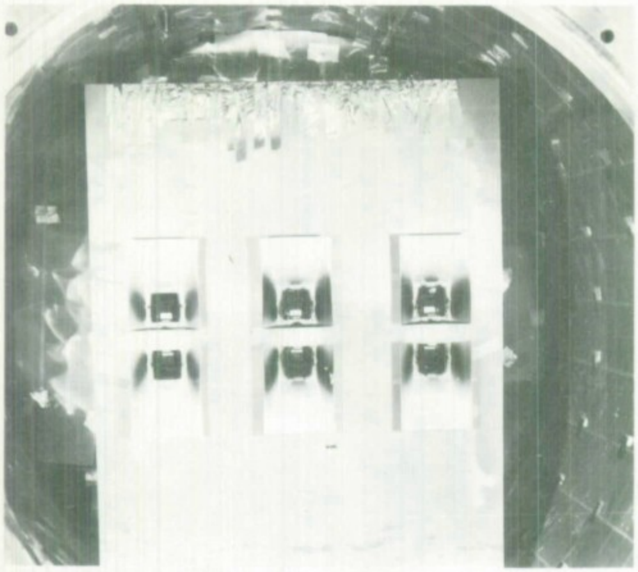

Fig 5. The Thruster Arrangement for the 3 PPT test. It is identical to that of Fig. 2.

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